CHAMBER TECHNOLOGY FOR SPACE-STORABLE PROPELLANTS

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INTRODUCTION AND SUMMARY

This paper summarizes the research efforts performed under NASA Contract NAS7-304, "Chamber Technology for Space Storable Propellants," during the period 1 July 1964 through 30 April 1965. This 12-month applied research program, which is still in progress, has as its objective the establishment of the technology necessary for the design of cooled thrust chambers capable of an 1800-second firing duration at a minimum c* efficiency of 95 percent, using the high performance oxidizer, oxygen difluoride, and a hydrazine-type fuel. The program is both analytical and experimental in nature.

The analytical studies were conducted to select the most promising fuel and chamber cooling technique, as well as the design of the test hardware. As a result of these analyses, monomethylhydrazine (MMH) emerged as the best choice, and a thrust chamber configuration was selected that consisted of an ablative combustion chamber, a regeneratively cooled throat section, and an ablative nozzle skirt. The analyses also defined the optimum chamber pressure and mixture ratio for the experimental phases of the program.

The experimental study (at the 1,000-pound thrust level) provided specific design criteria for a cooled thrust chamber using the selected propellant combination. Several candidate injector designs were evaluated in a brief series of experiments (which include a demonstration of throttling capability), and a self-impinging doublet pattern was found to be the most compatible with the selected chamber cooling concept. The other thrust chamber components (ablative chamber, regeneratively cooled throat, and ablative skirt) were then fabricated and evaluated under sea level and simulated altitude conditions. The results indicated that all of the cooled thrust chamber components could be expected to maintain their structural integrity for the required 1800-second duration while providing a c* efficiency of at least 95 percent. Finally, in a series of 1800-second firings (presently under way), the feasibility and performance of the selected chamber cooling concept are being demonstrated at sea level and altitude conditions.

ANALYTICAL STUDIES AND PRELIMINARY DESIGN

The fuel selection and thrust chamber cooling analyses, which were interrelated, led to the selection of the specific fuel and the specific thrust chamber cooling concept (Fig. 1). The various thrust chamber components used in the subsequent experimental work were designed in accordance with the results of these analyses.

FUEL SELECTION ANALYSIS

Eleven representative fuel candidates were chosen from among the hydrazine and amine-class fuels. Four neat fuels were included because of their simplicity and well-characterized physical properties, as well as the extensive development experience associated with them. Three binary blends (which represent simple mixtures with improved space storage capability and increased thermal stability), and four ternary blends (which represent the ultimate in present-day tailor-blending techniques), were also included. Ammonia was considered to be hypergolic with oxygen difluoride in this analysis, even though its hypergolicity under the subject engine operating conditions is somewhat doubtful. The 11 candidate fuels are shown in Table I.

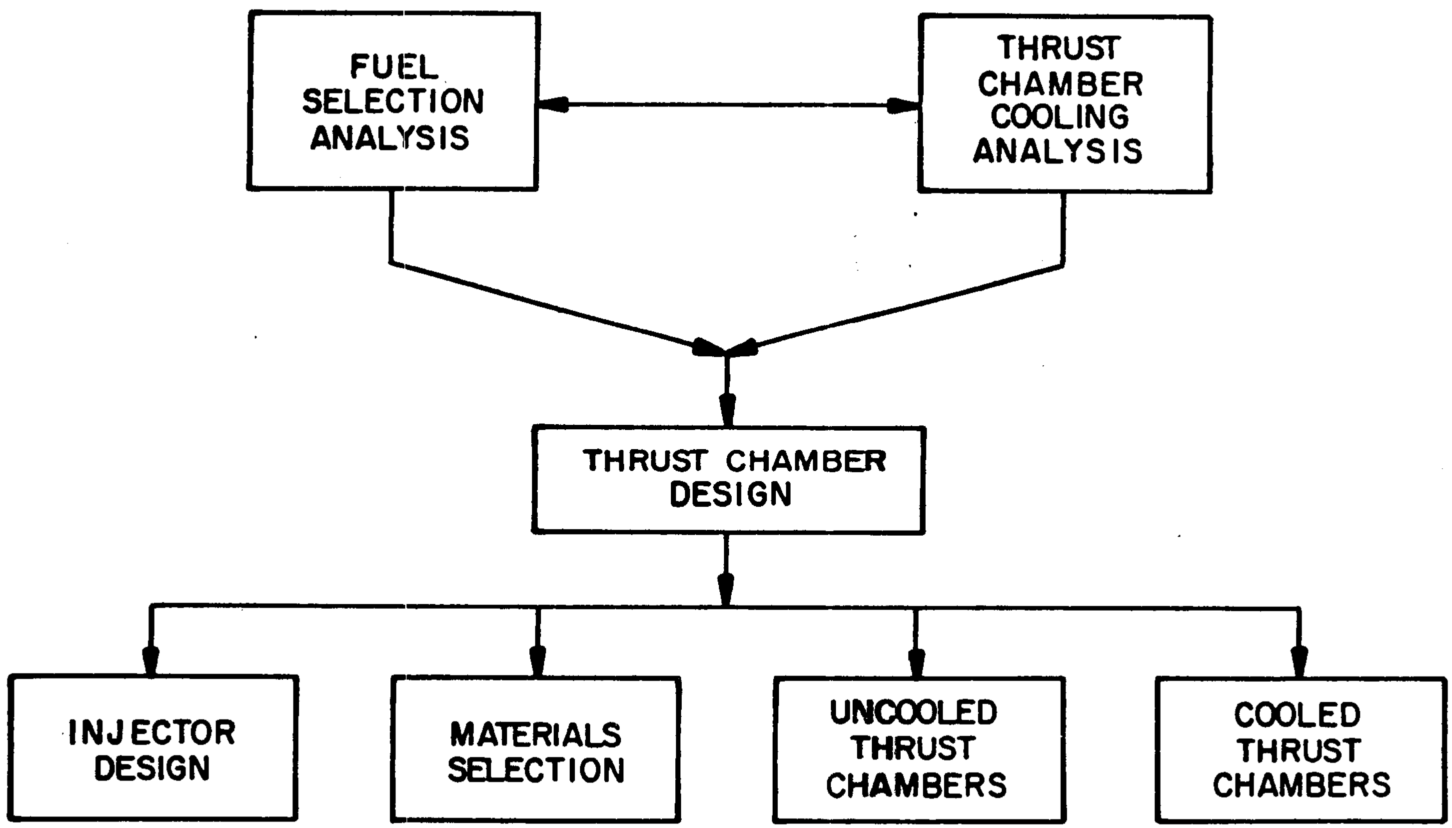


Figure 1. Technical Action Flow Diagram for Analysis and Design

TABLE I

CANDIDATE FUELS

(Selected From Hydrazine and Amine Class
Neat Fuels and Blends)

PRIMARY NEAT FUELS	BLENDS
HYDRAZINE	50/50
UDMH	MHF-3
MMH	MAF-4
NH_3	MHF-5
	MAF-1
	HYDRAZOID-P
	BA-1014

These candidate fuels were then compared in terms of performance, payload, and space storage capability with oxygen difluoride, as well as operational aspects and thrust chamber cooling capability. The fuels were rated as "strong" or "weak" candidates in each comparison area; from the high-ranking fuels in each area, the best compromise candidate was finally selected.

Performance

The theoretical specific impulse for each of the 11 candidates with oxygen difluoride (for shifting equilibrium) was calculated, and the results are summarized in Fig. 2. It should be observed that there is no great difference in the theoretical performance attainable with any of these fuels (the maximum spread is only about 15 seconds of impulse). However, there is a rather widespread in optimum mixture ratio, ranging from about 1.5 for Hydrazoid-P to about 3.0 for MAF-1 and MAF-4. The latter is an important design parameter, since it may influence tank size and regenerative thrust chamber design.

No single fuel could be rated as "strong" or "weak" based solely on these performance comparisons, to which little significance can be given without simultaneously considering payload and storage capabilities.

Payload and Space Storage

Analytical studies were accordingly made to establish a fuel rating based on the relative payload and space storage capabilities of each fuel. The approach was to select several probable spacecraft models and several likely space storage durations, and calculate the payload for each spacecraft-duration combination with each of the 11 candidates. For each model and duration, the fuel with the highest payload was rated at 100 percent; for the other fuels, the percent of maximum payload was calculated, and a weighted average obtained for each fuel based on all nine spacecraft-duration combinations.

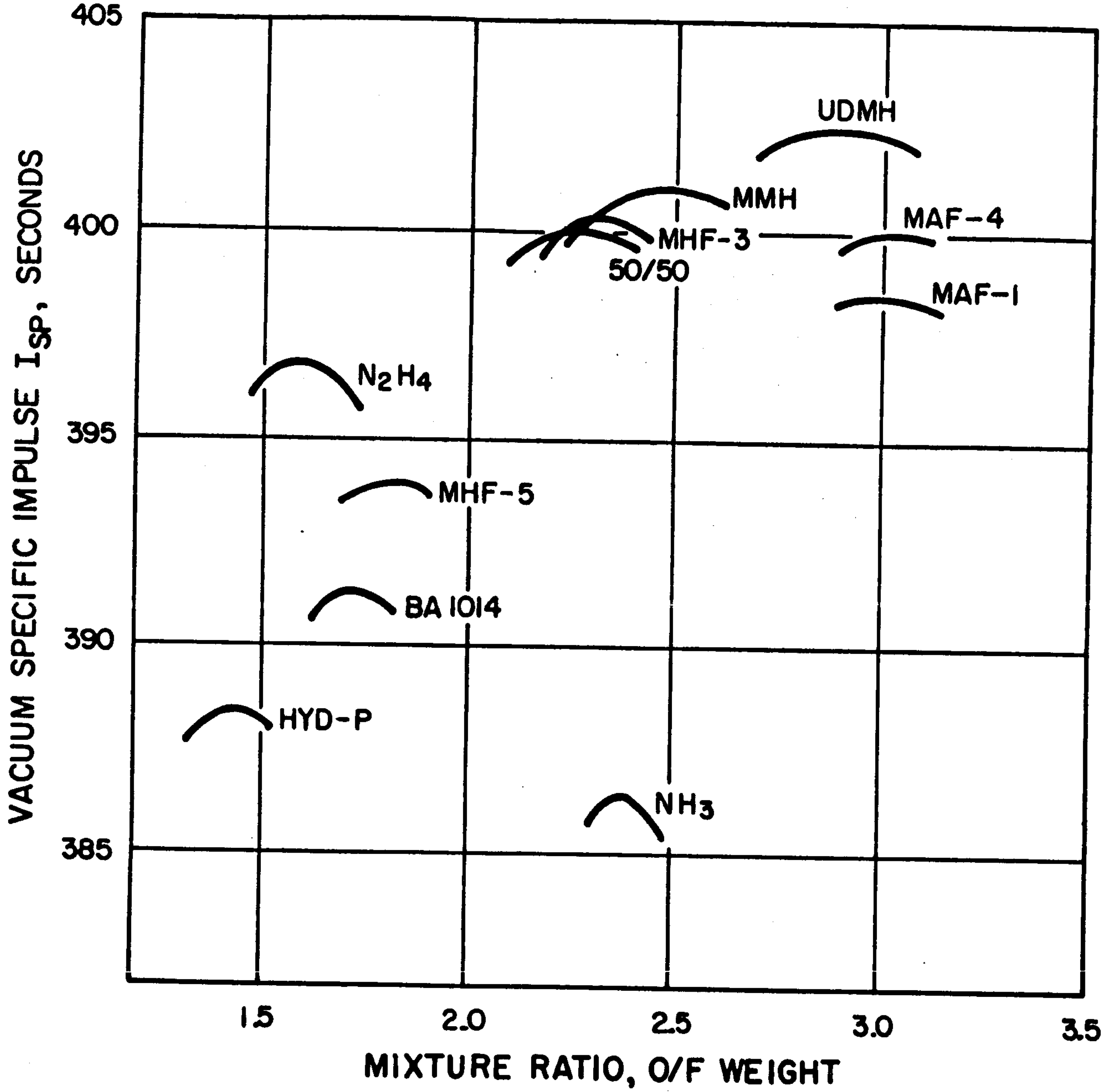


Figure 2. Theoretical Shifting I_{sp} for Eleven Candidate Fuels With OF_2 (ϵ = 40, P_c = 100 psia)

The spacecraft models selected were:

1. The Mars (MEM) ascent stage (29,500-pound propellant), which employs propellant tanks exposed to the space environment;
2. The Apollo Service Module (35,000-pound propellant); and
3. The Apollo Descent Stage (9,000-pound propellant), both of which feature tanks enclosed within the vehicle

The propellant weights of these vehicles bracket the range of potential applications of oxygen difluoride systems in the 1,000 to 30,000 pound thrust range.

The space storage durations considered were:

1. Short duration (10 days or less)
2. Medium duration (about 1 year)
3. Extended duration (about 2 years)

The results of the combined payload and space storage rating analysis are shown in Table II. This table shows that five fuels rate as "strong" candidates, and that, with the exception of MAF-1, they are all neat fuels or simple binary blends.

Operational Aspects

This rating area was intended to include other factors of importance in the development of an engine system which exhibit a variation depending on the fuel. The factors evaluated were: (1) experience, both with the fuel and with the appropriate propulsion systems; (2) the relative simplicity of systems development, including the sensitivity of the fuel's density and viscosity to temperature; and (3) propellant logistics in terms of ease of handling and commercial availability. Each area was separately

TABLE II

PAYLOAD AND SPACE STORAGE RATING RESULTS

FUEL	RATING	
MMH	72.1	"STRONG"
MAF-4	71.8	
UDMH	71.6	
MHF-3	71.3	
MAF-1	70.3	
NH_3	65.1	
HYDRAZOID-P	52.9	"WEAK"
MHF-5	52.0	
BA-1014	37.3	
50-50	16.5	
N_2H_4	16.4	

analyzed and the candidates rated from 0 to 10. In the final combined rating, each category was weighted nearly equally, with a slight emphasis placed on experience.

Table III presents the final results of the operational aspects ratings. These results should not be construed to precisely differentiate between fuels separated by only a few points, but a separation into "strong" and "weak" groups can definitely be made. The five "strong" candidates are all neat fuels or simple binary blends.

System Optimization Analysis

To determine the optimum values of several critical parameters which influence chamber cooling (chamber pressure, mixture ratio, and nozzle expansion area ratio), as well as the maximum allowable deviations of these parameters from the optimum values for a 2-percent payload reduction, a system optimization analysis was conducted. It was based on the selected propulsion models already mentioned, and considered perturbations in mission, vehicle, and propulsion system characteristics. The results of this study are summarized in Table IV, and show that rather wide variations may be tolerated in these design parameters with a relatively small loss in payload.

THRUST CHAMBER COOLING ANALYSIS

The thrust chamber cooling analysis consisted of: (1) a comprehensive qualitative review of possible cooling methods, (2) a detailed quantitative analysis of gas-side heat transfer, and (3) evaluations of a number of cooling techniques (including combinations) for each of the fuel candidates. The cooling analysis determined the best fuels from a cooling standpoint and also the most promising cooling methods for the 1,000- to 30,000-pound-thrust range.

TABLE III

OPERATIONAL ASPECTS RATING

FUEL	FINAL RATING	
NH_3	84	"STRONG"
MMH	78	"STRONG"
UDMH	74	"STRONG"
N_2H_4	73	"STRONG"
50-50	67	"STRONG"
MAF-4	62	
MHF-3	54	"WEAK"
MAF-1	53	"WEAK"
BA-1014	44	"WEAK"
MHF-5	42	"WEAK"
HYD-P	40	"WEAK"

TABLE IV

RESULTS OF SYSTEM OPTIMIZATION ANALYSIS

PARAMETER	OPTIMUM VALUE AND MAXIMUM ALLOWABLE DEVIATION *
CHAMBER PRESSURE	100 PSIA ± 50 %
MIXTURE RATIO	MAX Isp VALUE ±25%
EXPANSION AREA RATIO	100:1 ± 50 %

* - BASED ON A 2 % PAYLOAD REDUCTION

For the sake of brevity, the results of the quantitative cooling analysis are presented only for four fuel candidates, MMH, UDMH, NH_3 and Hydrazoid-P. MMH and UDMH are presented because they represent the strongest overall candidates based on performance, payload, space storage, and operational aspects. Although NH_3 was neither the strongest nor the weakest in the overall ratings, it is presented here because it is a neat fuel with an exceptionally high operations rating, and might potentially be used with relative ease in an engine system. Hydrazoid-P is included because it is a superior regenerative coolant with reasonably good space storage characteristics, even though its overall rating, exclusive of cooling, is weak.

As a result of the qualitative review, regenerative, radiation, ablation, and film cooling, as well as combinations of these, were selected for the detailed quantitative analyses. Both cooling techniques and fuel coolants were evaluated at standardized operating conditions and three thrust chamber sizes to permit an objective comparison. The conditions were:

Thrust level	1000, 4000, 30,000 pounds
Chamber pressure	100 psia
Mixture ratio (o/f)	value at -2 percent of maximum payload based on Bray performance
Expansion area ratio	40:1 (77 percent bell)
Contraction area ratio	3:1
Combustion chamber cylindrical length	4 inches
Contraction half angle	30 degrees

The analysis also considered the compatibility of each cooling method with throttling over about 10:1 thrust ratio range at altitude.

Detailed calculations of chamber and nozzle heat flux, total heat load, and film coefficients were made for each of the candidate fuels. The simplified Bartz equation was used (with the Eckert reference temperature and an axisymmetric velocity based on local wall Mach number) to predict the convective contribution; to this were added the contributions of gaseous radiation and boundary layer chemical recombination. In general, the analytically determined throat heat fluxes and total chamber heat loads were comparable for all the propellant combinations investigated. Typical values are presented in Table V.

Cooling Technique Evaluations

To evaluate the regenerative cooling technique and the regenerative cooling capability of the candidate fuels, both heat sink capability and relative coolant-jacket pressure drops were analytically investigated. The maximum heat sink capability was determined at each of the three thrust levels (1000, 4000, and 30,000 pounds) and then compared to the integrated chamber heat load based on the gas-side heat transfer calculations. The results, which are summarized in Fig. 3, show that only Hydrazoid-P could completely regeneratively cool the engine at either 30,000 or 4000 pounds, and that complete regenerative cooling is almost possible with MMH in a 30,000-pound-thrust engine. However, by utilizing a radiation-cooled skirt from 40:1 down to the minimum area ratio attach point (for radiation cooling), it was found possible to cool thrust chambers down to the 1000-pound level with Hydrazoid-P, and down to somewhat less than the 4000-pound level with MMH. With the other candidate fuels, it was not possible to cool thrust chambers below the 30,000-pound-thrust level, even with this combined regenerative-radiative technique.

The velocity head ($\rho V^2/2 g_c$) required for regenerative cooling in the throat region was calculated for each of the candidate fuels, since this is a convenient index of the relative coolant jacket pressure drops. These velocity heads are compared in Fig. 4, which indicates the higher pressure drops to be expected with UDMH and Hydrazoid-P compared to NH_3 and MMH.

TABLE V

TYPICAL CALCULATED VALUES OF HEAT FLUX AND HEAT LOAD FOR OF_2 WITH HYDRAZINE-TYPE FUELS

THRUST LEVEL LB/F	THROAT HEAT FLUX, BTU/IN^2 SEC	TOTAL CHAMBER HEAT LOAD, BTU/SEC
1000	6.3	530
4000	5.5	1570
30,000	4.5	8700

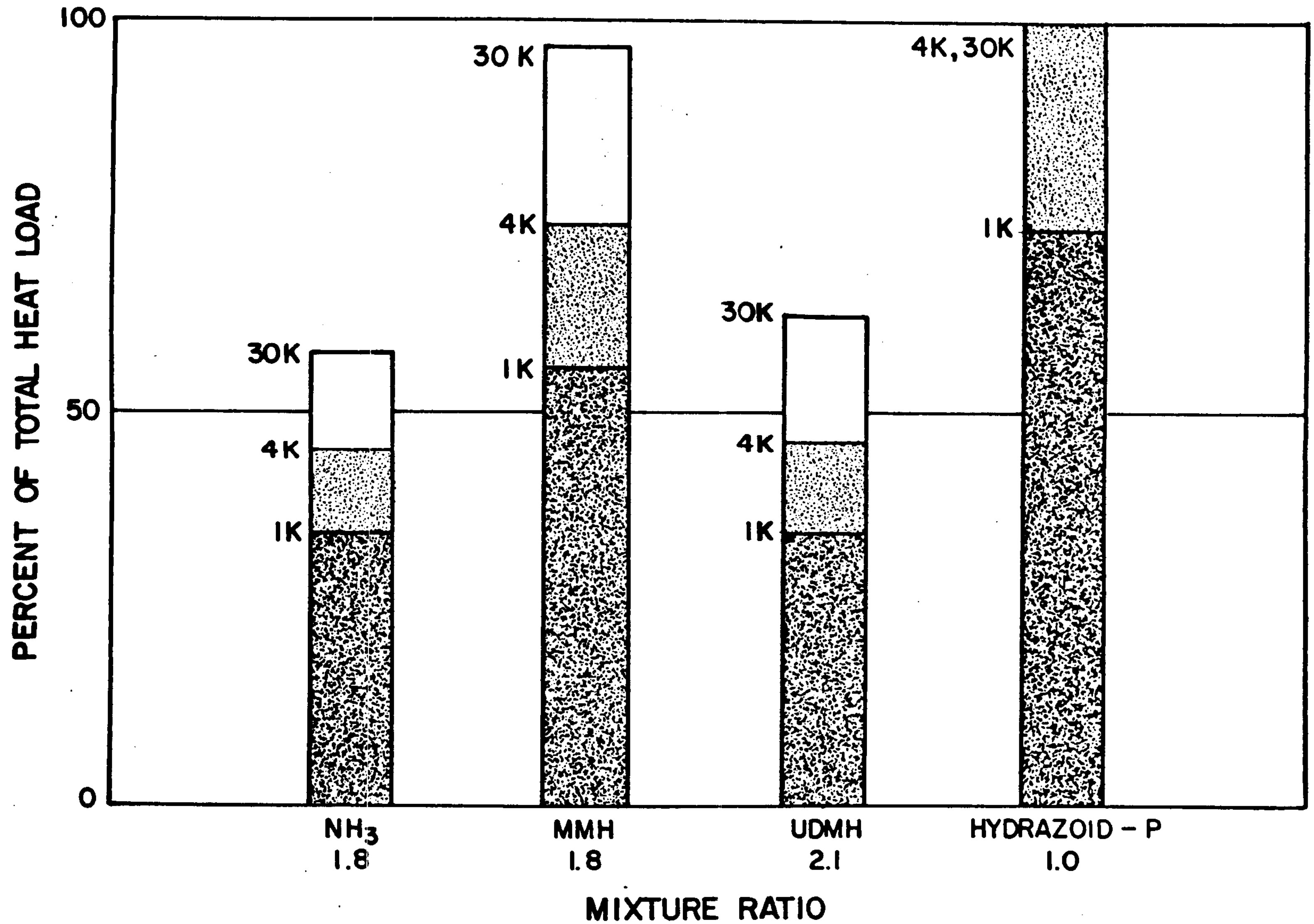


Figure 3. Calculated Total Heat Sink Capability for Complete Regenerative Cooling for Three Chamber Sizes and Four Candidate Fuels Based on Gas Side Heat Transfer With Oxygen Difluoride Oxidizer

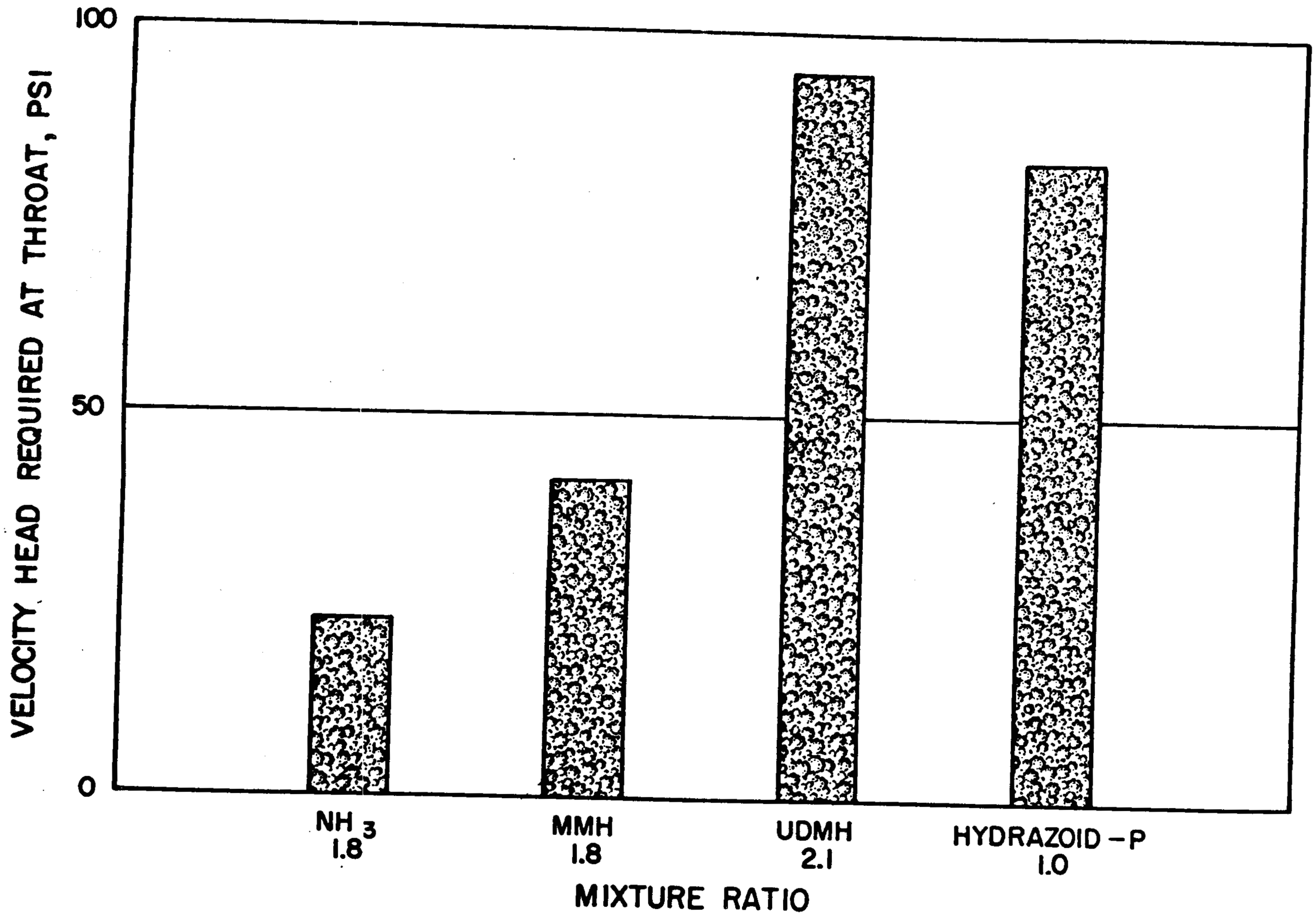


Figure 4. Relative Coolant Pressure Drop

It was recognized that a regeneratively cooled thrust chamber could only be throttled over a very limited range, because of the possibility of thermal decomposition of the fuel or local burnout phenomena at the lower fuel flowrates encountered during throttling. Since little or no throttling is normally required in space engine applications at the 1000-pound thrust level (the scale of the present experimental thrust chambers), it was not considered to be a serious disadvantage. At higher thrust levels (30,000 pounds, for example), however, deep throttling requirements could seriously limit the applicability of a regeneratively cooled thrust chamber.

Although the successful use of ablative materials for the throat region was originally questionable for long firing durations (because of probable throat enlargement), application of ablative materials to the combustion chamber was considered. Earlier experimental thrust chamber firings at Rocketdyne with another propellant system showed negligible surface erosion with phenolic carbon cloth ablative material, which was as thermodynamically predicted for combustion gases containing a small amount of water vapor. Although the water content for NH_3 and Hydrazoid-P are relatively high (greater than 10 weight percent), the water contents of MMH and UDMH are low and can be made to approach zero by decreasing the mixture ratio (Fig. 5). It was recognized that this could be achieved by a suitable injector design which gave a low mixture ratio in the vicinity of the chamber wall.

Based on extrapolations of data from previous ablative firings under similar conditions, it was predicted that the char depth could be held to about 2 inches after 1800 seconds of firing by using a carbon cloth phenolic material (which resists erosion in the absence of water vapor in the exhaust gases), backed up by phenolic refrasil (which chars at about half the rate of the carbon cloth).

Through the use of such a composite ablative material it should be feasible to completely cool a 1000 pound OF_2/MMH thrust chamber. The ablative material could be used for the straight cylindrical portion of the combustion chamber, while regenerative cooling could be used in the throat region. A radiation or ablation-cooled skirt would be used for the remainder of the expansion.

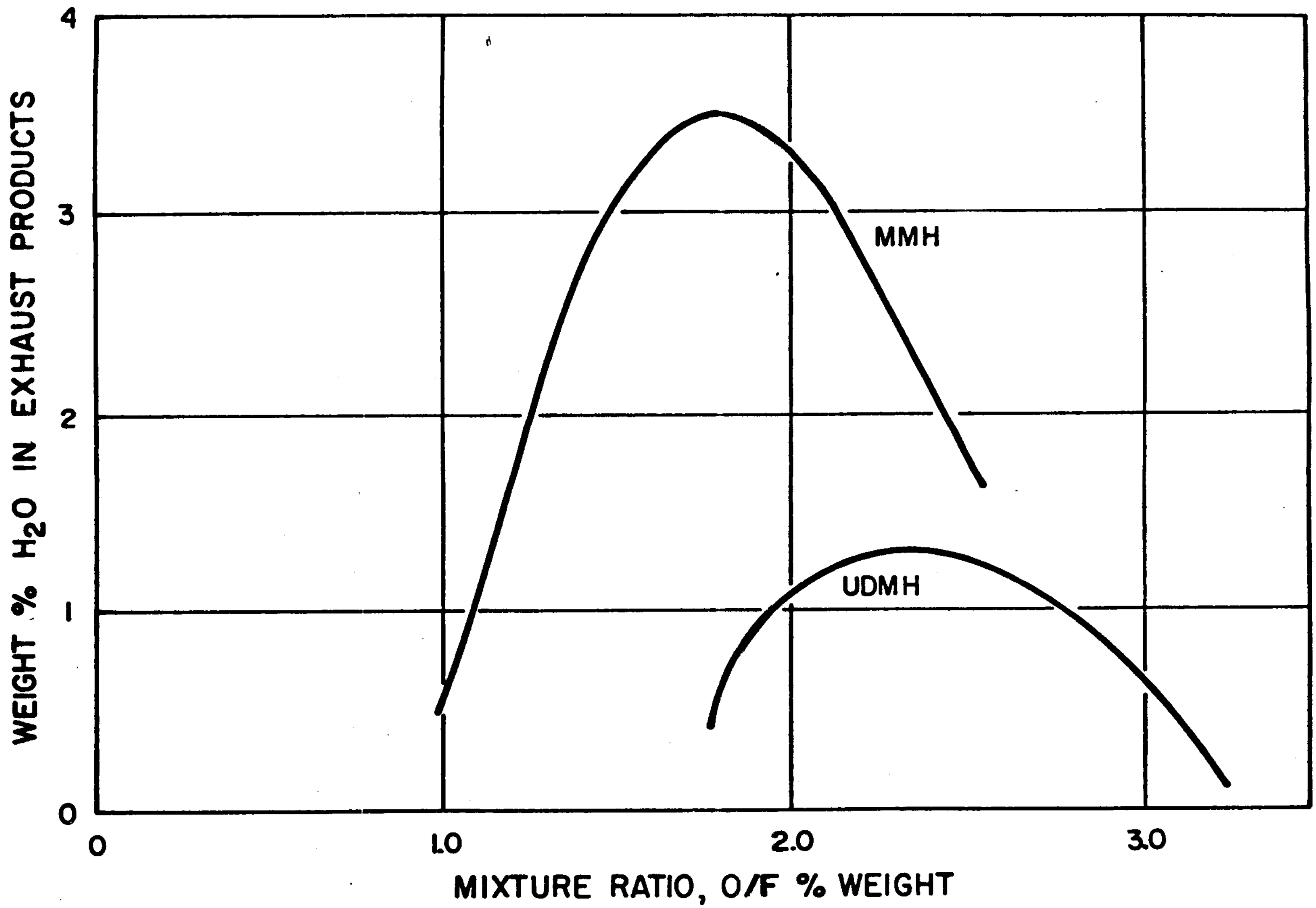


Figure 5. Percentage of Water in Combustion Gas Products as a Function of Mixture Ratio for Two Candidate Fuels With OF_2

Furthermore, an ablation-cooled thrust chamber is capable of throttling over a relatively wide thrust range, provided that mixture ratio gradient control can be maintained by means of a suitable injector.

Film cooling was considered as an alternative to ablation at the 1000-pound thrust level; the level which was selected for experimental evaluations under this program. Calculations of coolant requirements were carried out using previously obtained experimental data on the fraction of the total coolant enthalpy which is actually effective in cooling a chamber wall. The results for all the fuels were similar: about 5 weight percent of the total propellants would be required to fuel film cool just the short cylindrical portion of the combustion chamber. Under these conditions, the injector c* efficiency would have to be greater than 100 percent to achieve the required 95 percent c* efficiency in the chamber. In addition, to cool longer chambers might require staged cooling, with a resulting increase in the mechanical complexity of the system. Finally, the use of film cooling would seriously limit the throttling capability of the present chamber, and might make accurate thrust chamber scaling difficult. It therefore did not appear that the use of film cooling would be as desirable as the use of ablation, at least in the present application.

FUEL SELECTION AND THRUST CHAMBER COOLING ANALYSIS SUMMARY

Of the four fuels discussed for the thrust chamber cooling analyses, no single fuel was found to be outstanding for universal application with the oxygen difluoride oxidizer. The cooling analyses indicated that a fully regeneratively cooled, a fully ablative-cooled, or a film-cooled thrust chamber were not feasible at the 1000 pound thrust level, and that the most promising concept was a composite thrust chamber consisting of an ablative-cooled combustion chamber, a regeneratively cooled throat section, and a radiation- or ablation-cooled skirt section.

The physical properties of Hydrazoid-P indicated that this fuel would be a good candidate for regenerative cooling. However, since a fully regeneratively cooled engine was found not to be feasible below a thrust level of

4000-pounds thrust, and Hydrazoid-P exhibited a weak overall rating as a fuel candidate, it appeared to be less desirable. In addition, it may not be compatible with any ablative chamber design approach, because of its relatively high exhaust gas water content. MMH, on the other hand, was found to be a very compatible fuel for regeneratively cooled configurations, and was also seen to be quite compatible with ablative configurations (provided the mixture ratio near the ablative walls could be controlled through suitable injector design), which made it a very good compromise choice. UDMH would have been a better choice if completely ablative chambers had been feasible; however, it was quite weak for regenerative applications.

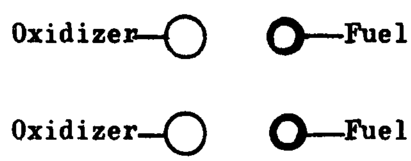
TEST HARDWARE DESIGN

Injector Design Philosophy

The design of injectors began relatively early during the program, before the composite thrust chamber concept just described had emerged as the most workable chamber cooling approach. Accordingly, injector design proceeded along two parallel paths; two injectors were designed with a view primarily toward the requirements of completely regeneratively cooled chambers, while a third was designed primarily for potential ablative-cooled chamber applications. The injector design constituted a relatively minor portion of the overall effort, because it was felt that high performance could readily be attained through the application of well-established injector design criteria for optimum propellant mixing and atomization. In addition, the task was markedly simplified by using considerable experimental information relative to halogenated oxidizers and hydrazine-type fuels, already on hand at Rocketdyne.

Although general injector design criteria and related experimental data may be applied to any injector, the approach chosen for the design of a specific injector depends to a large extent on the thrust chamber cooling technique. Therefore, some of the injector design considerations for regeneratively and ablative-cooled chambers will be briefly summarized here.

Injectors intended primarily for regenerative applications are normally required to produce maximum performance in a relatively short combustion chamber, since long chambers may lead to high jacket pressure drops or an excessive coolant bulk temperature rise. The latter effect must be avoided, since it may result in exothermic decomposition, unstable film boiling and burnout, or a change of phase of the coolant. Consequently, an injector capable of producing finely atomized, uniformly distributed sprays as soon as possible after the propellant streams leave the orifices, is required. In general, unlike-impingement orifice patterns may be expected to produce a higher degree of propellant mixing than their self-impinging counterparts (for the same degree of atomization), since the fuel and oxidizer impinge directly, and therefore mix sooner. Therefore, the two injectors designed for potential regenerative applications were both of the unlike-impinging stream type. One was an unlike doublet, and the other was an unsymmetrical 2-on-2 with individual elements arranged as shown in the sketch below. Both contained 68 elements arranged in a square or "box" pattern.



The 2-on-2 and the unlike doublet injectors were used for the throttling capability demonstration tests which are discussed later in the paper.

Injectors intended primarily for ablative thrust chamber applications are normally required to have their orifice patterns oriented in such a manner as to prevent or at least minimize impingement of oxidizer-rich fans directly on the chamber walls, since this effect may cause severe local erosion. In addition, and even more important in the present case, the overall erosion of the ablative surface, which results mainly from thermochemical reactions at the surface, must be avoided.

These thermochemical reactions may be divided for convenience into two kinds: the reaction of water, and the reaction of uncombusted oxidizer, with the hot, carbonaceous char. It is clear that the overall erosion rate might then be held within tolerable limits by minimizing the quantities of water vapor and unconsumed oxidizer in contact with the wall. It was recognized that this could be accomplished by an injector which could maintain low mixture ratios in the wall region (recall how water vapor content decreases sharply with mixture ratio for OF_2/MMH). Such an injector must also have a relatively steep mixture ratio gradient between the optimum performance value in the bulk of the free stream, and the reduced value in a small zone near the wall, to avoid performance degradation. In general, radially oriented self-impinging orifice patterns have been found to be most effective in maintaining such a controlled mixture ratio gradient, as well as in preventing localized erosion. Accordingly, the injector designed for potential ablative applications was of this type. Since the thrust chamber cooling analysis subsequently indicated the desirability of using an ablative combustion chamber, the self-impinging doublet injector will be emphasized over the other two candidates in this paper.

Self-Impinging Doublet Injector

The self-impinging doublet injector has 80 elements arranged in a radial pattern of alternating fuel and oxidizer rings (Fig. 6). The outermost ring consists of fuel elements (pairs of fuel orifices, each 0.020 inch in diameter) canted 10 degrees inward to avoid direction impingement of the propellant fans on the chamber walls. The next ring toward the center consists of oxidizer elements (pairs of oxidizer orifices, each 0.025 inch in diameter), also canted 10 degrees inward to prevent direct wall impingement. Equal oxidizer and fuel pressure drops were employed at the flowrates required to maintain a chamber pressure of 100 psia at a 2.0 mixture ratio.

The orifice pattern in Fig. 6 was designed to produce a gradient in mixture ratio across the face of the injector, from fuel rich conditions (mixture

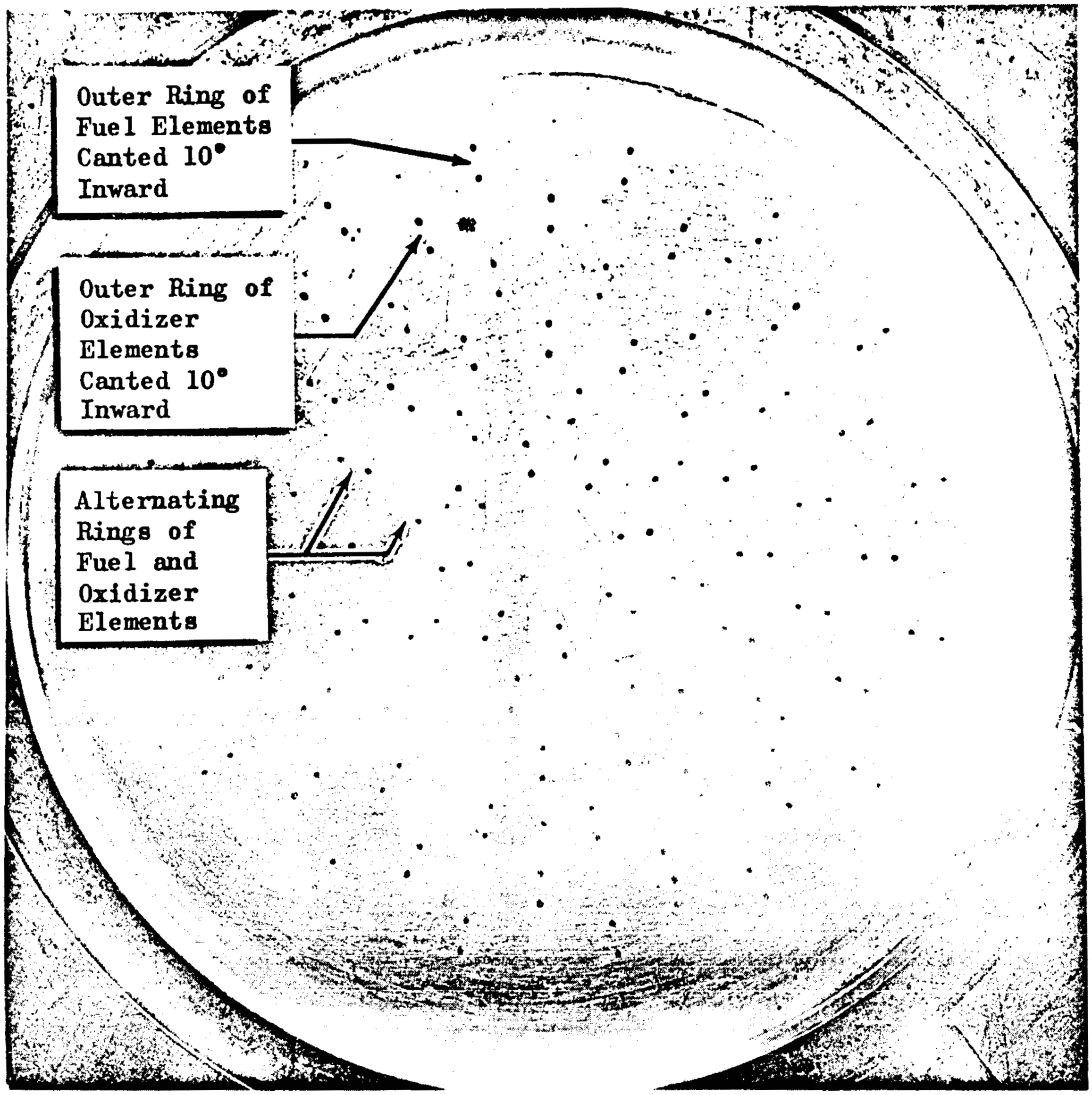


Figure 6. Orifice Pattern of Self-Impinging Doublet Injector RID 2381

ratios of 1.0 or less) at the outer periphery to the nominal design mixture ratio (2.0) in the central portion of the spray.

To determine that each injector would produce a sufficiently uniform degree of propellant mixing, and that the self-impinging doublet injector produced in addition the desired mixture ratio gradient control, a brief series of cold-flow spray analyses was conducted prior to the hot-firing evaluations.

All three injectors were cold flowed into a spray-sampling device, using the propellant simulants water and trichlorethylene at a nominal mixture ratio of 1.8 to 2.0, with the injector face at a constant distance from the collector. The mass and mixture ratio distributions were then calculated by a computer program.

The results of the cold flow experiments indicated that all three injectors could be expected to produce the required 95 percent c* efficiency in a chamber with a characteristic length of 15 inches or less. In addition, the desired mixture ratio gradient was attained with the self-impinging coublet injector, (Fig. 7).

Figure 7 is a normalized profile plot of the mixture ratio distribution for the self-impinging doublet injector. It may be seen that the central portion of the spray pattern, comprising 95 percent of the total mass of the spray, had a nearly uniform mixture ratio of about 2. In the vicinity of the walls, a sharp and nearly linear mixture ratio gradient existed, from 2 to less than 1 at the walls. Only about 5 percent of the total mass of the spray was collected in the gradient regions. This injector was therefore expected to help maintain the structural integrity of the thrust chamber during long-duration firings.

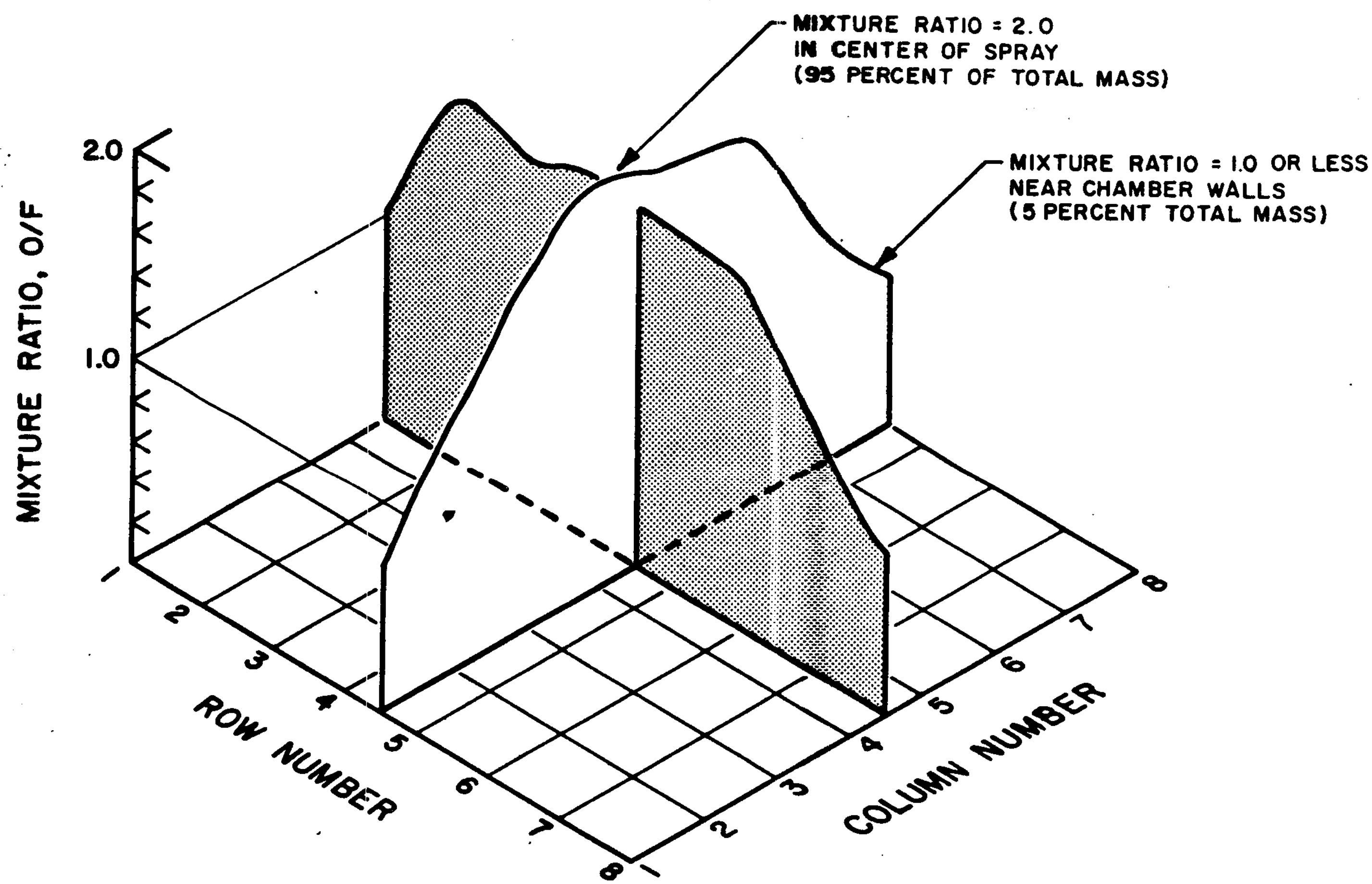


Figure 7. Mixture Ratio Distribution in Spray of Self-Impinging Doublet Injector

Ablative Chamber Liners

The results of earlier Rocketdyne research showed that certain combinations of commercially available ablative materials were capable of withstanding the combustion chamber environmental conditions (such as temperature, pressure, and chemical composition) encountered with interhalogen oxidizers and hydrazine-based fuels at chamber pressures of 300 psia or less. Methods of combining two materials (an inner liner of carbon cloth and an outer layer of Refrasil to maintain a low overall rate of char), to yield a chamber assembly which has a distinct potential for long-duration firings, had already been demonstrated.

Accordingly, a detailed experimental survey of a number of candidate ablative materials was not considered necessary, and only the physical arrangement of the two materials (to provide both structural integrity and a long-duration capability) was investigated during this program. The liners were all composites of carbon cloth and Refrasil phenolic materials, and differed only in the thickness of the carbon cloth layer and the angle (measured from the chamber centerline) at which the carbon cloth layer was wrapped. The angle of wrap affects the char rate of the carbon cloth material, as well as the thermal conductivity of its char (these properties are not isotropic), and the thickness of the carbon cloth affects its temperature profile. Both variables also influence the rate of char of the Refrasil overwrap. In addition, the wrap angle will determine to a large extent the ability to withstand the shear forces of the accelerating gas. Carbon cloth thicknesses of 0.6 and 1.2 inches, and wrap angles of 15 and 30 degrees were studied. The Refrasil was wrapped parallel to the chamber centerline in each case.

All the liners were 8 inches long (this length was based on the injector evaluation tests to be discussed later) and had an internal diameter of 3.72 inches. Each was instrumented with thermocouples embedded at various depths in both the carbon cloth and Refrasil, as one means of measuring the char rates of these materials.

Ablative Nozzle Skirts

Unlike the rather severe internal environment encountered in the combustion chamber and throat region of the thrust chamber, conditions in the nozzle expansion zone appeared to be quite amenable to the use of contemporary ablative materials for cooling. The results of the ablative firings conducted during this program (to be presented later on) indicated that the carbon-cloth/Refrasil composite material just described was quite suitable to use in the combustion chambers; there was no reason to believe, then, that the same material would not perform adequately under the less stringent conditions prevailing in the nozzle expansion cone. To conserve program costs, therefore, only the carbon cloth/Refrasil composite was evaluated as a skirt material. This represented a departure from the original program plan, which called for investigation of at least three different ablative materials. Also, less emphasis was placed on varying the layer thickness and wrap angle of the carbon cloth material, since the nozzle environmental conditions were expected to be much less severe than those in the chamber.

Each skirt consisted of an inner layer of carbon cloth/phenolic, 0.6-inch thick, and an overwrap of refrasil/phenolic. The skirts were conical, with a divergence half-angle of 15 degrees, and expanded the exhaust gases from an area ratio of 1.9:1 to one of 20:1. The skirts have not been evaluated as yet, but when they are fired, a supersonic diffuser will be used to ensure that experimental analysis of the skirt is not invalidated by flow separation. The skirts will also be instrumented with embedded thermocouples to measure the char rates of both ablative materials.

Regeneratively-Cooled Throat Section

The regeneratively cooled throat section for application with the ablative chambers and the ablative nozzle skirts has been designed, and several have been fabricated.

The final design selection (Fig. 8) was based on the following requirements: (1) ability to remove the local heat flux, (2) material compatibility with the coolant (monomethylhydrazine), (3) minimal pressure drop through the coolant passages, and (4) ease and cost of fabrication.

The design selected consisted of a spiral slot machined into the coolant side of the nozzle contour, and appropriate filler sections to direct the coolant flow. Selection of the one-pass spiral design was dictated by stringent limitations on the mass flowrate of fuel, particularly at the low experimental thrust level. Design values of film coefficients were obtained from the heat flux data measured in the uncooled hardware evaluations (to be described in the following section).

Material selection was limited primarily to stainless steel and the aluminum alloys because of their compatibility with the MMH coolant. Type 321 stainless steel was chosen as the preliminary material for the nozzle contour while aluminum alloy nozzle inserts were fabricated to allow for materials evaluation in this particular application.

In addition, a water-cooled throat section was used in some of the early ablative firings to simulate the regeneratively cooled throat while it was in fabrication.

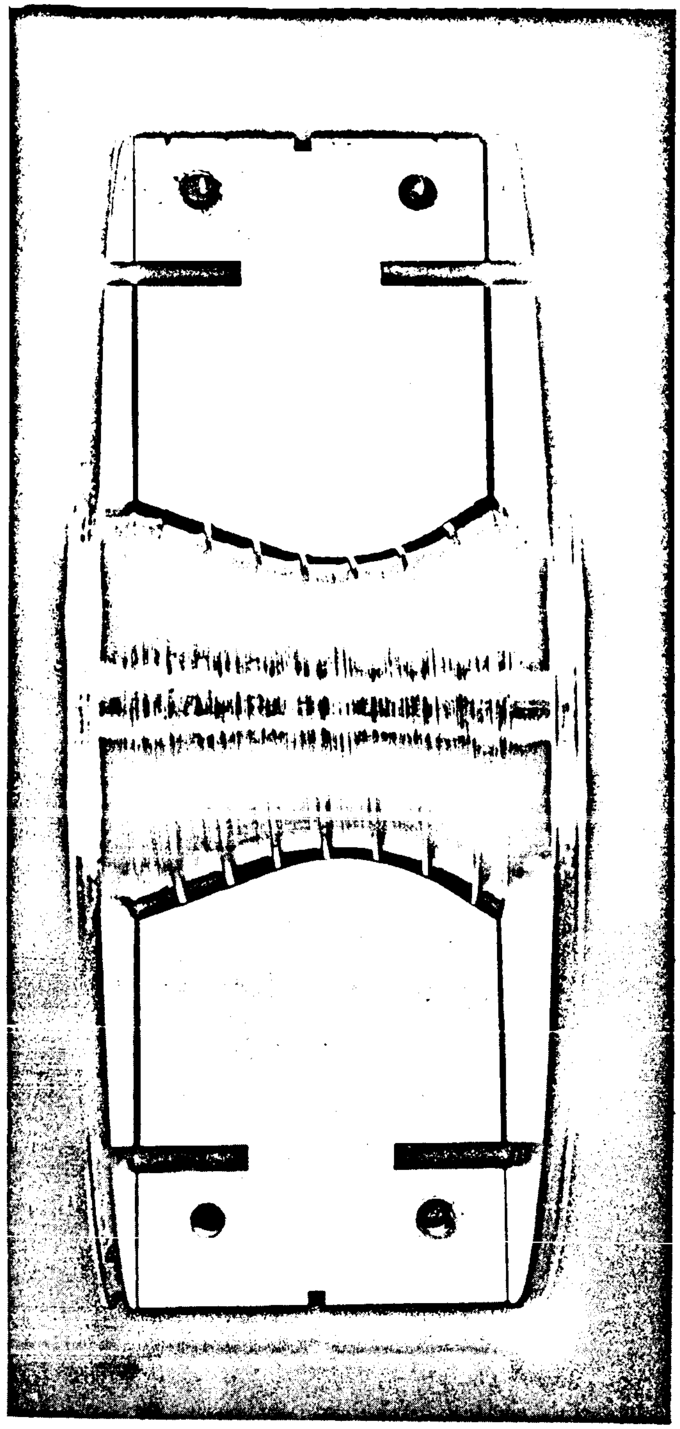


Figure 8. Cutaway View of Regeneratively-Cooled Throat Section

EXPERIMENTAL PROGRAM

The experimental program was conducted to obtain the design criteria necessary to build a long-duration cooled space engine using an oxygen difluoride oxidizer. As shown in Fig. 9, it consisted of (1) a brief series of injector evaluation firings to verify the ability of each of the three injectors to produce the required 95-percent c* efficiency in a relatively short combustion chamber, (2) cooled thrust chamber tests, in which several different ablative chamber and regenerative nozzle configurations, as well as injector materials of construction and ablative skirts, were studied, and (3) a series of start-stop-restart firings (totaling 1800 seconds for each thrust chamber assembly tested) to demonstrate the feasibility of the final cooled-chamber design under both sea level and simulated altitude conditions. To conserve program costs, most of the firings were made with 70-30 FLOX as a simulant for oxygen difluoride.

INJECTOR EVALUATION FIRINGS

The objective of this portion of the program was to document c* performance as a function of characteristic chamber length (L*) for each of the candidate injectors. Chamber pressure and mixture ratio were held at nominal values of 100 psia and 2.0, respectively, in the majority of tests, in accordance with the results of the Task I performance optimization analysis. Transient chamber and nozzle heat flux measurements were made simultaneously, using the uncooled copper combustion chamber and nozzle shown in Fig. 10.

Test Hardware

Figure 10 is a schematic representation of the engine hardware used in the injector evaluation tests. Combustion chambers with 3.72-inch diameter and of variable length were fabricated, so that characteristic chamber length L* could be systematically varied between 7.73 and 20.19 inches (length varied

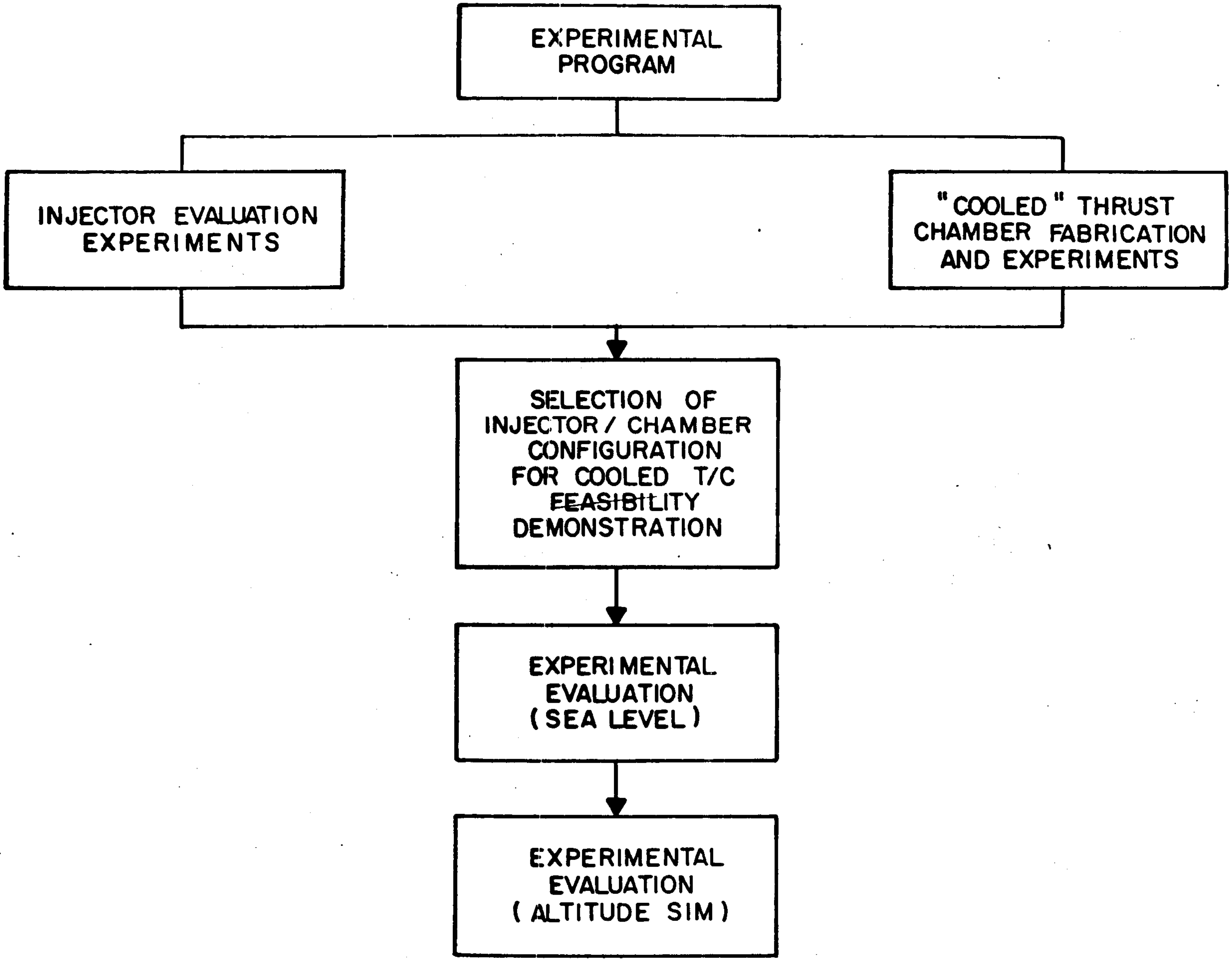


Figure 9. Technical Action Flow Diagram for Experimental Program

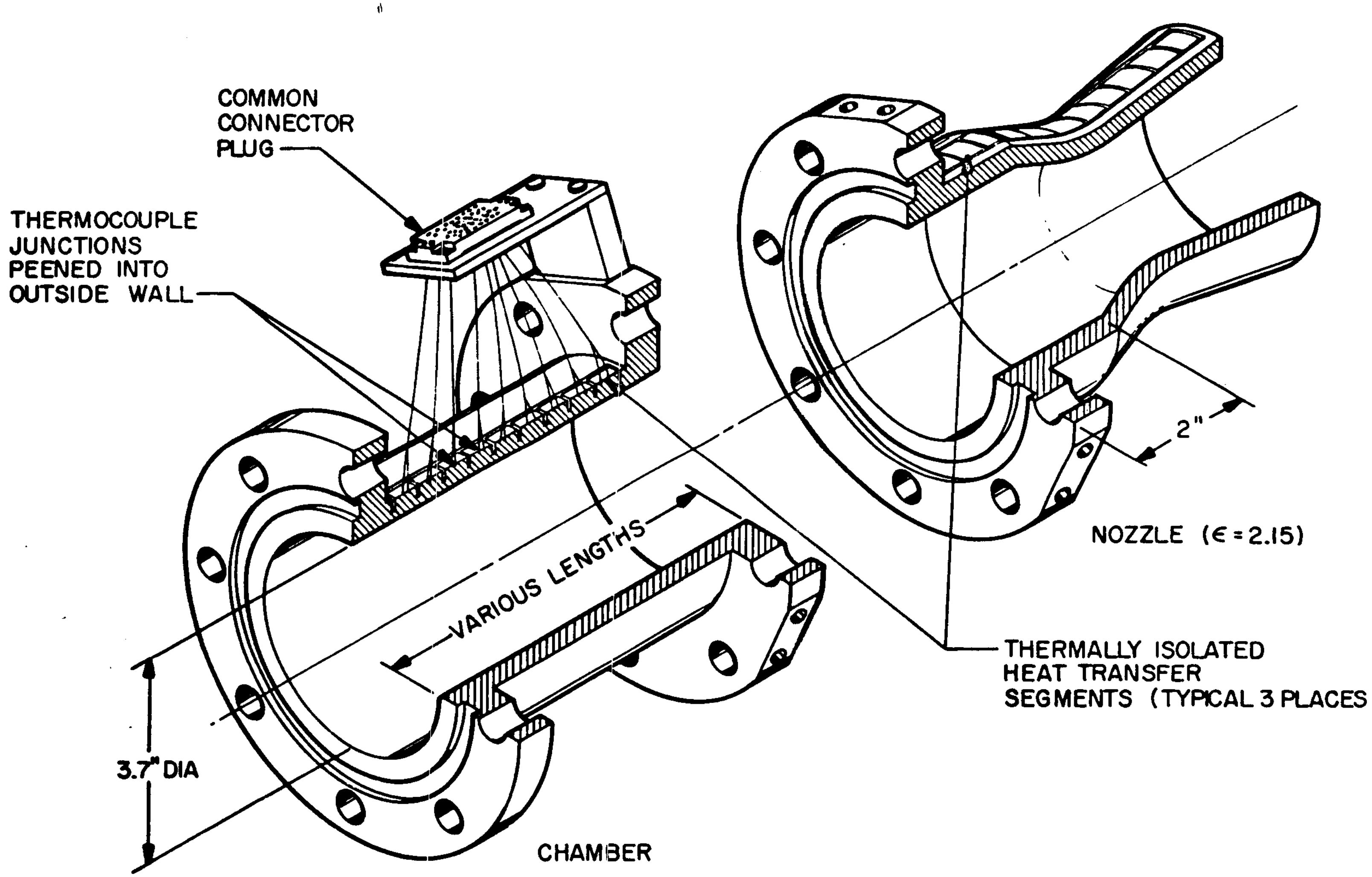


Figure 10. Schematic Representation of Thermally-Instrumented Uncooled Motor for Injector Evaluation Studies

between 3.22 and 9.47 inches). The nozzle section had a 2-inch-long cylindrical portion to permit injector evaluations to be carried out at very short chamber lengths where the variation of performance with the injector spray parameters was quite pronounced. A contraction ratio of 2.15 was employed, and expansion was optimum to the atmospheric pressure at the elevation of the Rocketdyne Propulsion Field Laboratory.

The chamber and nozzle sections were fabricated of copper. They were lined with a thin gold plating to prevent the reaction of amine fuel with the copper. Each piece had three rows of thermally isolated heat transfer segments, 120 degrees apart, formed by milling 1/32-inch slots to half the wall thickness. Into the outer surface of each slot was peened a chromel-alumel thermocouple junction. The thermocouple wires from each section terminated in a common connector plug to facilitate patching the temperature-response instrumentation into the Beckman 210 Data Acquisition System. Local values of the transient heat flux and film coefficients were then calculated by means of a computer program.

Fifty-one injector evaluation firings were conducted, 41 using 70-30 FLOX (to simulate oxygen difluoride) and monomethylhydrazine, and 10 using oxygen difluoride and monomethylhydrazine. Firing durations ranged between 1.5 and 3.0 seconds.

Injector Performance

Figure 11 shows the variation of experimental c* efficiency with chamber length for the self-impinging doublet injector. The circles represent FLOX/MMH firings, whereas the triangles represent OF_2/MMH firings. The values of c* efficiency shown have been corrected for heat losses to the copper chambers.

The performance efficiency with OF_2/MMH is seen to be slightly higher than with FLOX/MMH. Efficiencies in excess of 95 percent may be obtained with OF_2/MMH in chambers only about 7 inches long (L* = 15.3 inches). The reason for this slightly higher performance with oxygen difluoride is not immediately apparent,

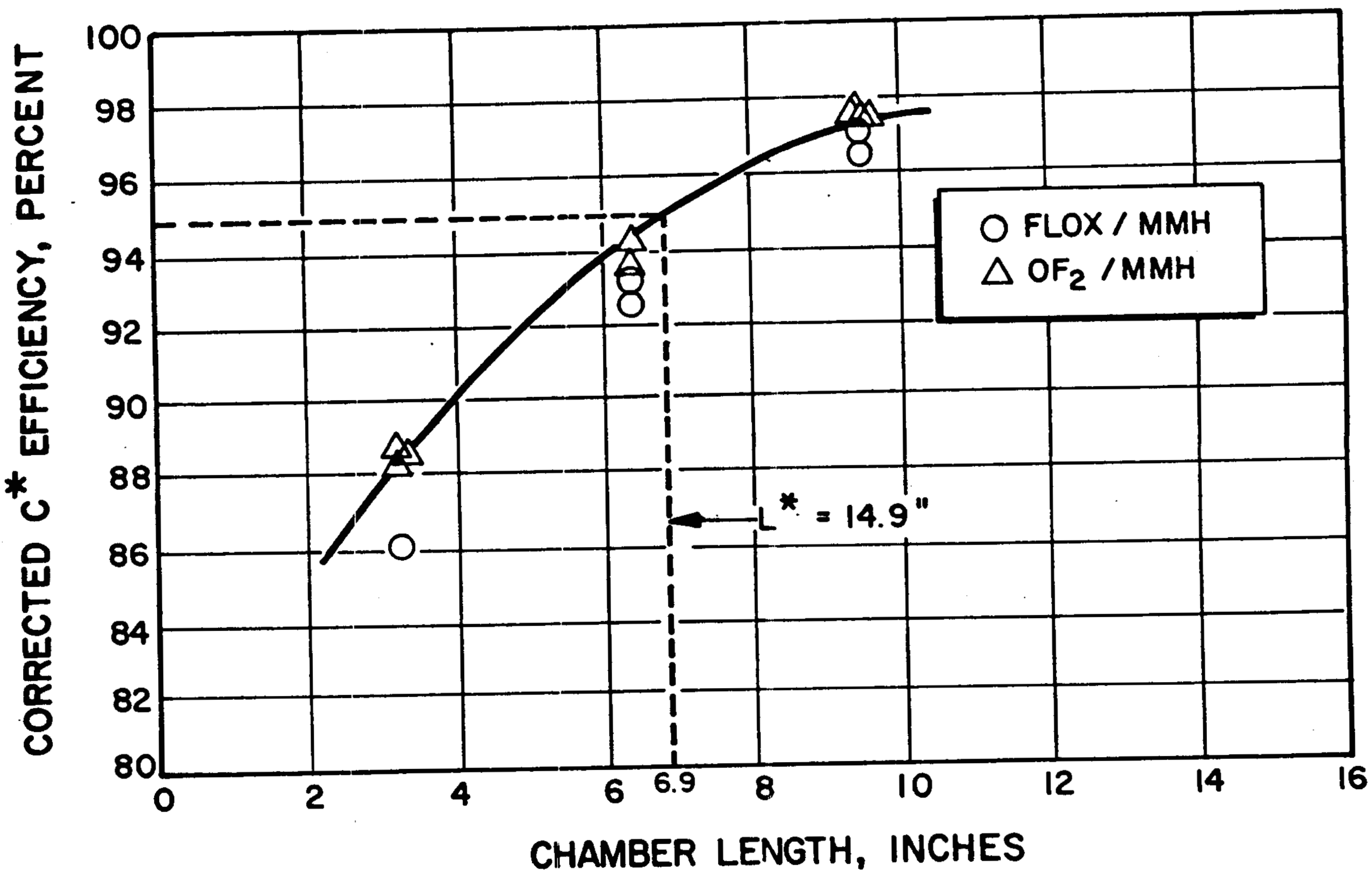


Figure 11. c* Efficiency vs Chamber Length for Self-Impinging Doublet Injector

although the trend is consistent. However, the precision of the c* measurement is not significantly better than the difference shown. These results indicate that 70-30 FLOX is a good c* performance simulant for oxygen difluoride.

Similar performance results were obtained with FLOX/MMH for the other two injectors studied, and these results are summarized in Table VI, where the chamber lengths required for 95-percent c* (based on the FLOX/MMH experiments) are compared.

Throttling Demonstration

As a corollary to the injector evaluation experiments, a series of firings was conducted to demonstrate the potential feasibility of throttling a cooled thrust chamber over a range of about 10:1 at altitude, by means of a technique incorporating a change in injector pattern during the throttling cycle.

The 2-on-2 injector configuration was particularly suitable for such a demonstration of throttling potential. Throttling is accomplished by first reducing propellant flowrates through upstream valving until pressure drops become too low for stable operation. At this point, propellant flow can be cut off from one oxidizer orifice and the diagonally opposite fuel orifice by appropriate valving and manifolding, thus converting the pattern to a simple unlike doublet with a reduced overall flow area which could be throttled still further.

Fifteen fixed-point throttling tests were conducted with the unsymmetrical 2-on-2 and the unlike doublet injectors to demonstrate the feasibility of throttling over a thrust ratio of about 10:1 at altitude. The unlike doublet injector used in these throttling tests was not the same as that used in the injector evaluation firings already described. It had the same number of elements as the unsymmetrical 2-on-2, and each element consisted of only half the number of orifices, thus simulating the pattern which would result from the 2-on-2 when half the orifices were cut off during an actual throttling.)

TABLE VI

COMPARISON OF PERFORMANCE REALIZED FROM 3 CANDIDATE INJECTOR DESIGNS

INJECTOR	CHARACTERISTIC LENGTH FOR 95 PERCENT C* EFFICIENCY, INCHES
SELF-IMPINGING DOUBLET	14.9
UNLIKE DOUBLET	14.3
UNSYMMETRICAL 2-ON-2	18.0

Seven tests were made with the 6.47-inch uncooled chamber, and eight additional tests were made with the 9.47-inch long uncooled chamber. All firings were made using FLOX/MMH at a nominal mixture ratio of 2.0. Durations ranged from 3 to 5 seconds. Transient heat flux measurements were made concurrently at the 9.47-inch chamber length ($L^* = 19$ inches) to document variations in local film coefficient throughout the throttling cycle. In a series of six experiments (four with the 6.47-inch chamber, $L^* = 13$ inches and five with the 9.47-inch chamber), the 2-on-2 injector was fired at progressively lower flowrates (commencing with those corresponding to a chamber pressure of about 100 psia) to simulate throttling. When the injection pressure drops became marginally low from the standpoint of flow control (at a chamber pressure of about 50 psia) the 2-on-2 injector was replaced by the unlike doublet injector having the same general face pattern. Starting at the thrust level which prevailed prior to the pattern changeover, the unlike doublet injector was then further throttled by taking advantage of the higher pressure drops available at the lower chamber pressures. Six tests were conducted with this injector; three with the 6.47-inch chamber, and three with the 9.47-inch chamber.

The variation of c* efficiency with chamber pressure over the entire throttling range is shown in Fig. 12. As would be expected, performance efficiencies were significantly higher over the entire throttling range when the longer chamber was used. For example, it would seem to be possible to throttle this pattern combination from 100 to about 15 psia without degrading c* efficiency below 90 percent in the 9.47-inch chamber.

Effect of Injector on Overall Film Coefficient

The injector type was observed to influence both the magnitude of the average chamber and throat film coefficients, and the local circumferential variation of these coefficients.

Figure 13 shows the variation of the experimentally measured film coefficient (h_g) with both longitudinal and circumferential position along the combustion chamber for the self-impinging doublet injector (using FLOX/MMH) at a chamber

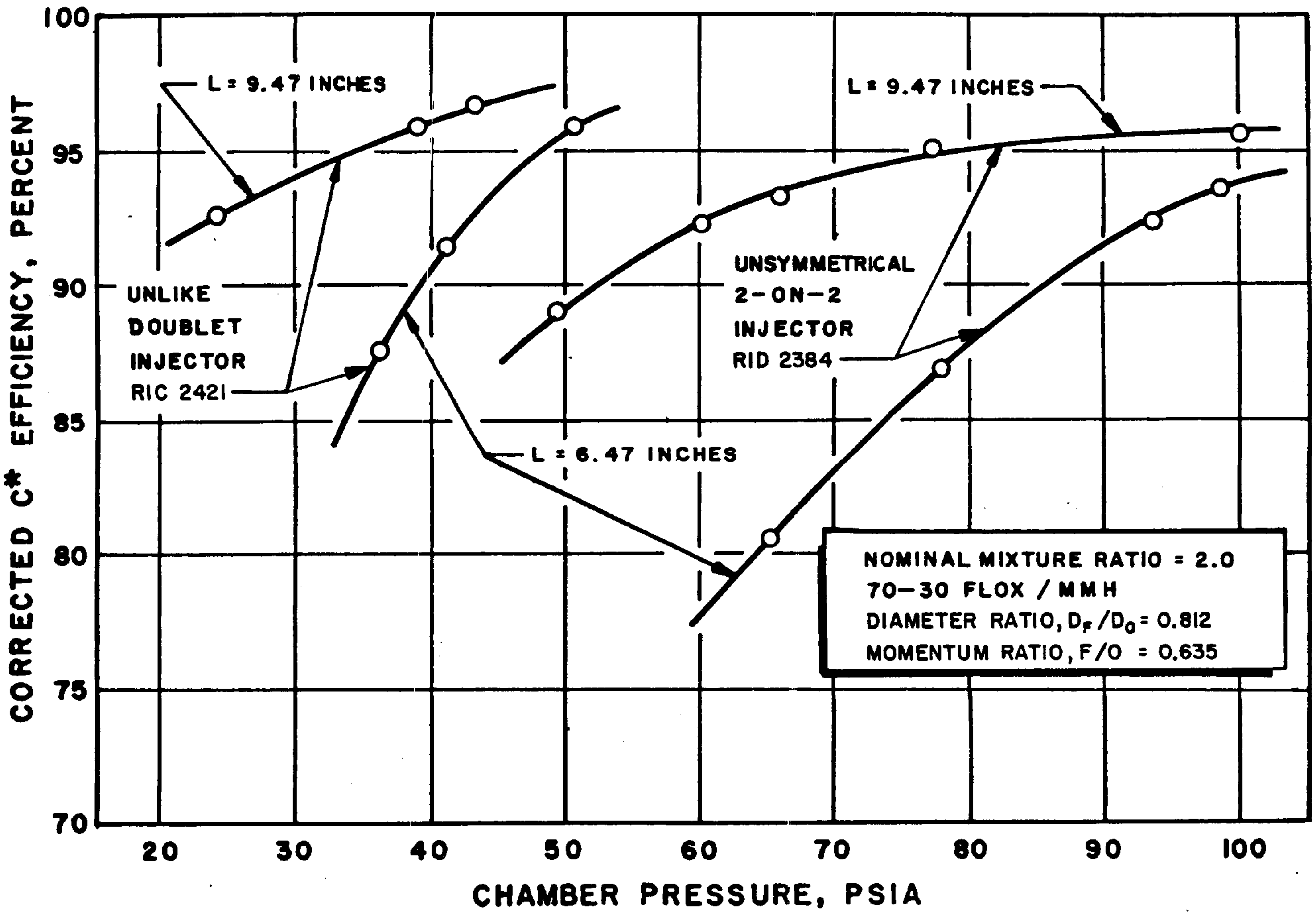


Figure 12. Variation of c* Efficiency With Chamber Pressure and Length for Sea-Level Throttling of the Unsymmetrical 2-on-2/Unlike Doublet Injector Combination.

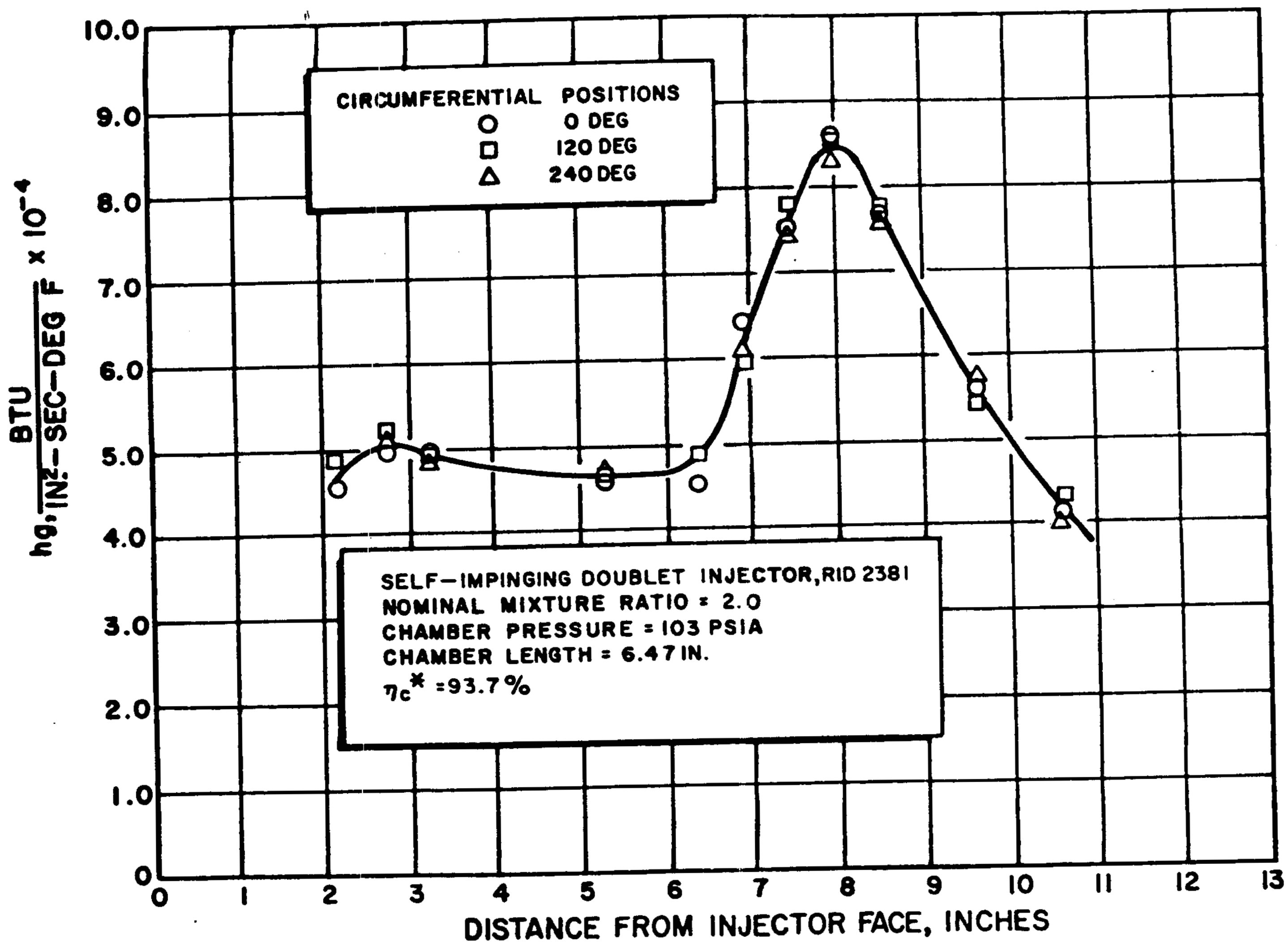


Figure 13. Typical Variation of Film Coefficient h_g With Longitudinal and Circumferential Location for the Self-Impinging Doublet Injector

pressure of 100 psia and a mixture ratio of 2.0. It is seen that the film coefficient is nearly constant at about 5×10^{-4} Btu/in.^{2}-sec F in the cylindrical portion of the chamber, and rises to about 8×10^{-4} just upstream of the physical throat.

Similar results were obtained for the other two injectors studies, and these results are summarized in Table VII, where the film coefficients are compared for all three injectors at a chamber length of 6-1/2 inches.

Effect of Injector on Local Variations in Film Coefficient

Figure 12 shows that there is negligible circumferential variation in the film coefficient with this particular injector design. Figure 14, however, which is a plot of the experimental data for the unlike doublet injector, shows deviations of up to ±10 percent from the average film coefficient in the chamber, and of up to about ±5 percent in the nozzle, for the unlike doublet injector. Similar deviations were found for the unsymmetrical 2-on-2 injector.

The lack of local circumferential variations in the case of the self-impinging doublet injector was the result of its radial pattern arrangement, and its canted outer rings of fuel and oxidizer elements, both of which prevented the direct impingement of oxidizer-rich propellant fans on the chamber walls. The other two injectors had orifice patterns arranged on square centers, and uniform wall inpingement was more difficult to achieve in that case. Local variations in heat flux and film coefficient were accordingly expected, and were correlatable with the injector orifice pattern layout.

Comparison of Oxygen Difluoride and FLOX Heat Transfer Data

The self-impinging doublet injector was fired with the oxygen difluoride oxidizer in the 6.47-inch and 9.47-inch chambers to verify both the performance

TABLE VII

EFFECT OF INJECTOR TYPE ON INDICATED CHAMBER
AND NOZZLE THROAT PEAK FILM COEFFICIENTS*

INJECTOR	FILM COEFFICIENTS, BTU / IN.^{2} - SEC - °F	
	CHAMBER	THROAT
SELF-IMPINGING DOUBLET	5.0×10^{-4}	8.0×10^{-4}
UNLIKE DOUBLET	5.5×10^{-4}	9.0×10^{-4}
UNSYMMETRICAL 2-ON-2	4.7×10^{-4}	9.5×10^{-4}

* NOTE: BASED ON 100 PSI CHAMBER PRESSURE
AND 6.47 INCH CHAMBER LENGTH

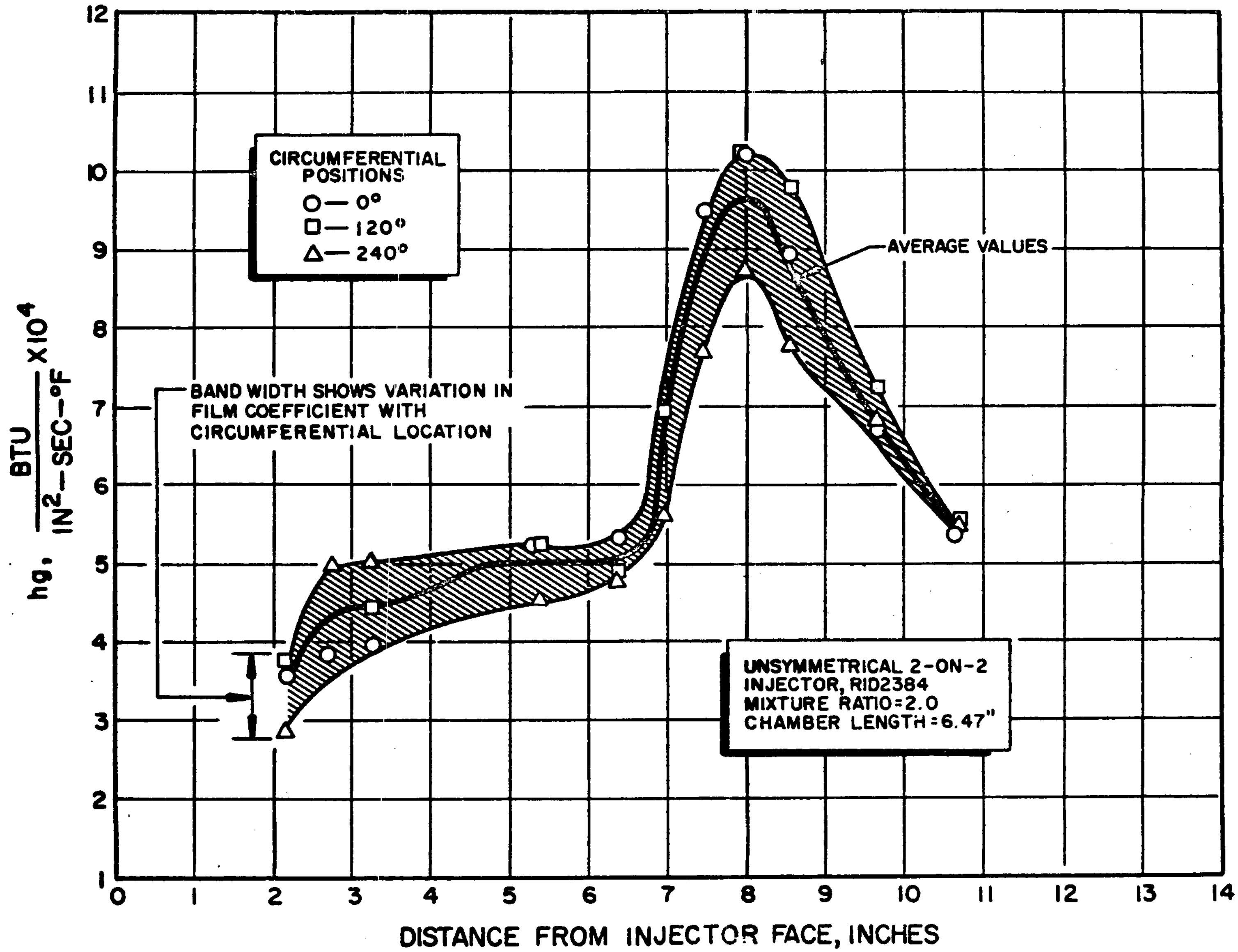


Figure 14. Typical Variation of Film Coefficient h_g With Longitudinal and Circumferential Location for the Unsymmetrical 2-on-2 Injector

and the heat transfer results obtained with the FLOX simulant. Values of local film coefficients obtained in the 6.47-inch chamber are shown in Fig. 15 for both the oxygen difluoride and FLOX oxidizers. All tests were conducted at a nominal mixture ratio of 2.0 and a chamber pressure of 100 psia. As seen from this figure, there is substantial agreement of the film coefficients when plotted against longitudinal position. Circumferential variations of the film coefficient obtained with oxygen difluoride were minimal, as was the case with the data obtained with the FLOX simulant with the self-impinging doublet injector. The results indicate that FLOX is a good heat transfer simulant for oxygen difluoride.

Comparison of Experimental and Theoretical Film Coefficients

A comparison of the experimental film coefficients with those predicted by the simple Bartz correlation at the same conditions is shown in Fig. 16. Physical properties for the Bartz correlation were based on frozen equilibrium. Good agreement between theoretical and experimental film coefficients is seen to exist at a wall temperature of 150 F. This temperature was the average wall temperature attained after about 2.0 seconds in the injector evaluation firings, all of which began with the engine hardware at liquid nitrogen temperature to prevent gassification of the FLOX (or oxygen difluoride) in the injector.

SELECTION OF INJECTOR PATTERN FOR USE IN COOLED THRUST CHAMBER EVALUATIONS

From the standpoint of realizable performance, all three candidate injectors were approximately equivalent, in that each was capable of providing a c* efficiency of 95 percent or greater in a reasonably short combustion chamber. However, because of the selected chamber cooling technique, it was necessary to choose that injector design which was potentially most compatible with an ablative chamber and nozzle.

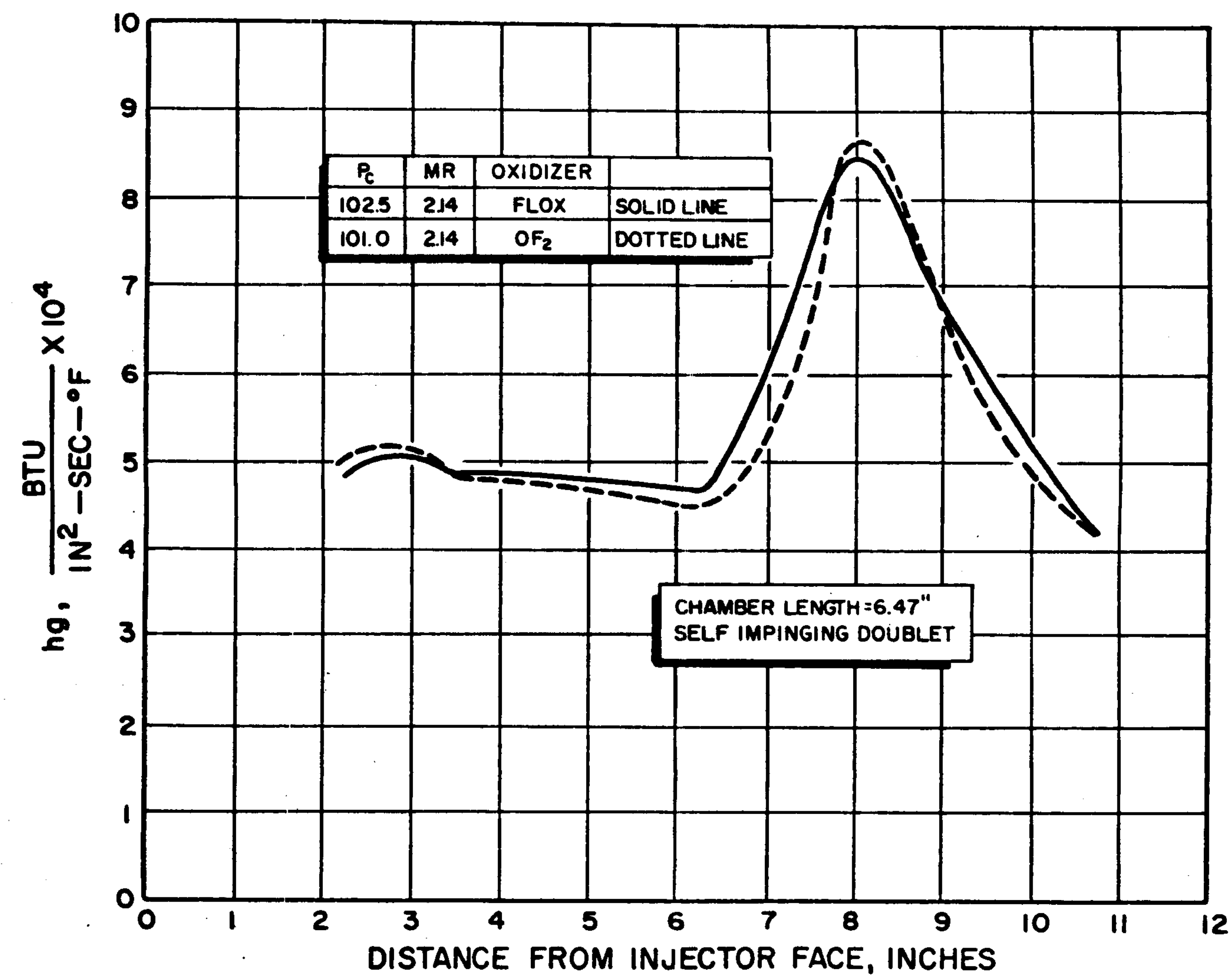


Figure 15. Comparison of Film Coefficients Realized With Self-Impinging Doublet Using FLOX/MMH and OF_2/MMH

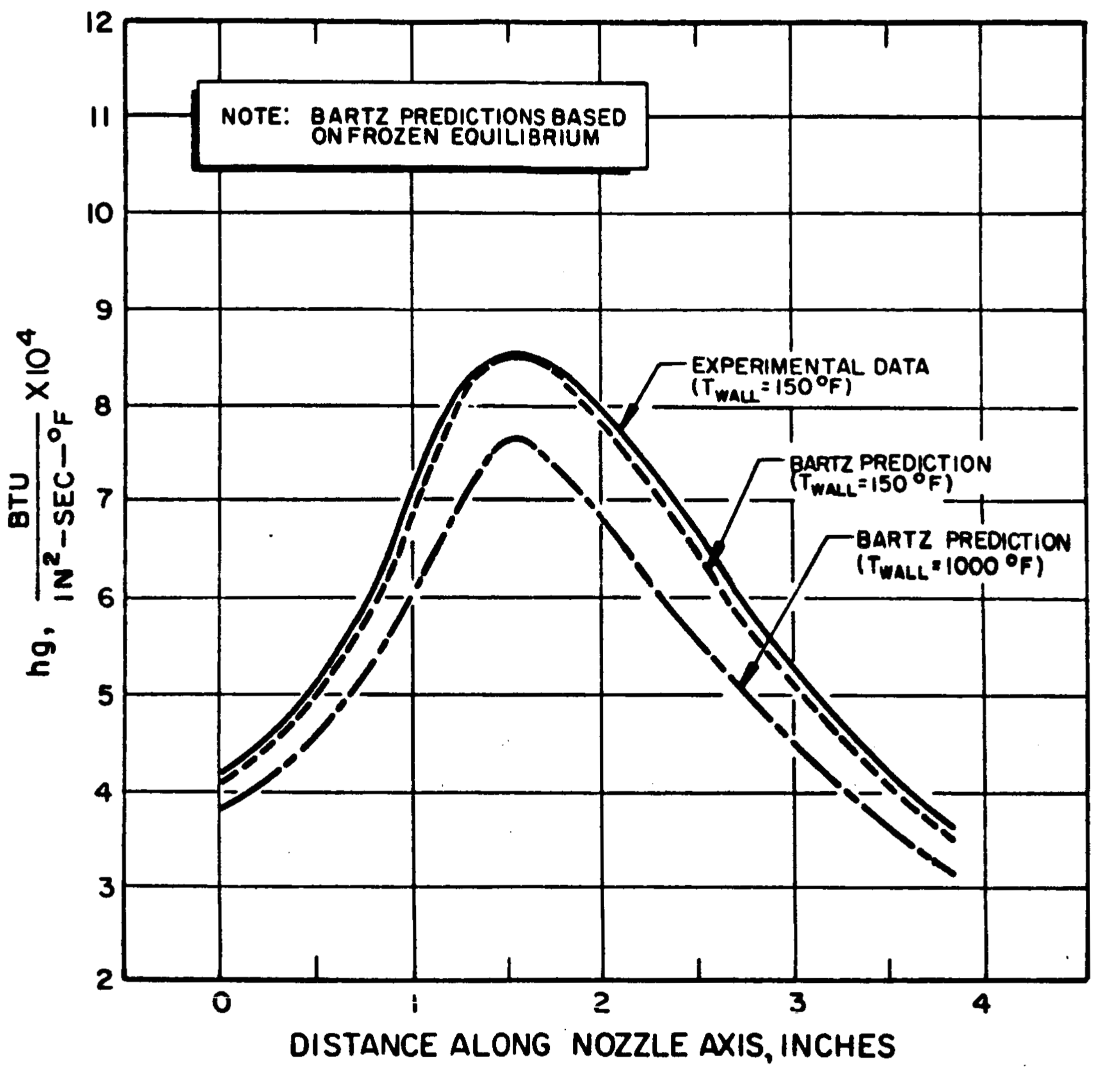


Figure 16. Comparison of Experimental and Theoretical Film Coefficients for a Gas-Side Wall Temperature of 150 F, and the Predicted Coefficient for a Wall Temperature of 1000 F

The self-impinging doublet injector was found to be superior to the unlike doublet and 2-on-2 injector patterns for the following reasons: (1) no circumferential heat flux variations, which, if present, could have influenced local char and surface erosion rates; and (2) its previously demonstrated ability to produce fuel-rich conditions near the chamber walls, which was expected to reduce the rate of surface regression by reducing the reaction of raw oxidizer with the char layer. Based on these considerations, the self-impinging pattern was selected for application in the cooled thrust chamber evaluations.

COOLED THRUST CHAMBER COMPONENT EVALUATIONS

As was shown earlier, the most feasible low-thrust-level cooling system appeared to be one which consists of an ablatively cooled combustion chamber, a regeneratively cooled throat region, and either a radiation or an ablative nozzle skirt. It is this engine configuration which is presently being used, along with the self-impinging doublet injector, to demonstrate the long-duration start-stop-restart capability, structural integrity, and performance level of the optimized thrust chamber design. Prior to the final feasibility demonstrations, at the engine assembly, a program was conducted to evaluate and "optimize" certain of the individually cooled components, such as injector materials, ablative chambers, ablative skirts, and regeneratively cooled throat sections. On all these tests 95-percent or higher c* efficiency was developed in the thrust chambers used.

Injector Material Performance

Several self-impinging doublet injectors were fabricated, one of 6061 aluminum alloy, one of type 321 stainless steel, and one of type A nickel, and all three were fired with FLOX/MMH. The aluminum injector showed signs of surface erosions and pitting in regions of high heat flux, and some enlargement of the MMH orifices was noted, after only 60 seconds of sustained firing. The stainless-steel injector showed similar signs of erosion and pitting in the high heat flux regions after 1000 seconds of firing, and the oxidizer injector orifice flow area was decreased slightly by deposits of iron oxide and fluoride.

These results indicate that aluminum, though chemically resistant and light in weight, may encounter face heating and erosion because of its inherently low melting point. Stainless steel, on the other hand, would possess better surface integrity at its higher melting temperature, but may be subject to chemical attack by the oxidizer at elevated temperatures. The ideal injector may be one possessing the durable features of both metals. Consequently, a nickel self-impinging injector is currently being evaluated, in the expectation that this metal may exhibit all of the good qualities of both the aluminum and stainless steel. To date, however, it has not been fired for long enough durations to permit a comparison with the other two metals.

Ablative Chamber Liner Performance

Two ablative chamber liners were fired using FLOX/MMH at a mixture ratio of 2 and a chamber pressure of 100 psia. Both liners had an inner layer of carbon cloth/phenolic oriented at 30 degrees to the chamber centerline, surrounded by parallel-wrapped Refrasil/phenolic. The chambers differed only in the thickness of the carbon cloth layers, which were 0.6 inch and 1.2 inches, respectively.

After each liner had been fired for 500 seconds (two 250-second tests each) core samples were removed from each liner and the char depths measured. The char depth at the midpoint of the 0.6-inch liner was 0.8 inch, indicating that the char front had passed into the Refrasil material. The char depth at the midpoint of the 1.2-inch liner was 0.85 inch, indicating that the char front remained in the carbon cloth layer.

Subsequently, the 0.6-inch liner was fired for an additional 850 seconds (in a series of 40-90 second tests), bringing the total duration on this chamber to 1350 seconds. At this point, the chamber was sectioned, and its appearance is shown in Fig. 17. The maximum char depth of 0.88 inch occurred at a point

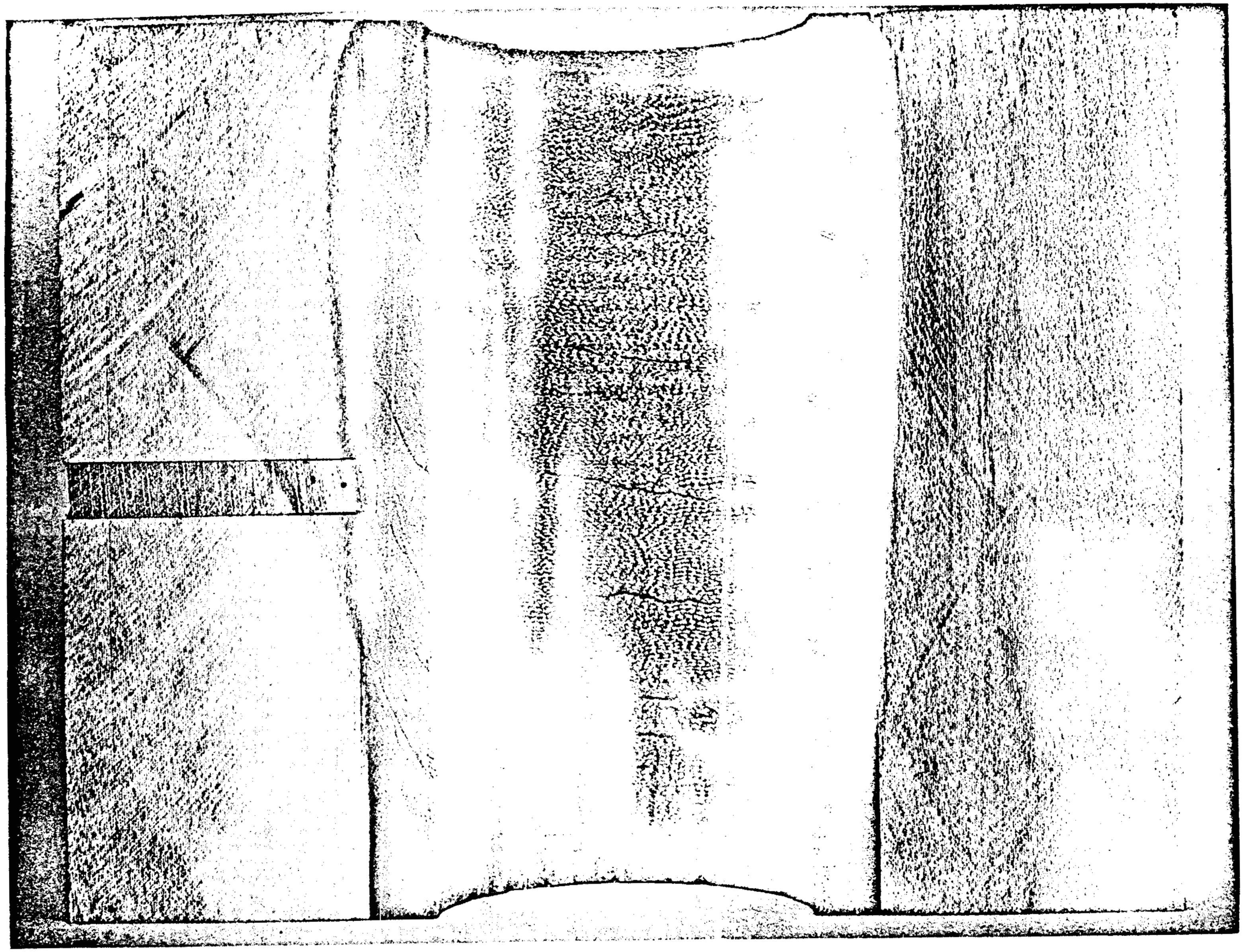


Figure 17. Sectioned Ablative Chamber After 1350 Seconds Accumulated Firing Duration

about 1 inch upstream of the chamber exit. At the injector end, where heat removal was provided by the cooled face of the water-cooled spacer, the char depth was only 0.2 inch (well within the carbon cloth layer).

The char depth varied only along the longitudinal axis of the chamber, and not circumferentially, indicating the effectiveness of the self-impinging doublet injector with mixture ratio gradient control. There were no deep gouges or "scallops," and the chamber was found to be structurally sound and suitable for additional firings. There was no evidence of surface erosion; on the contrary, the inside diameter of the chamber actually decreased from 3.72 inches to 3.66 inches over the 1350-second firing cycle.

In general, it appears from the limited quantity of experimental data gathered to date that the ablative liners evaluated so far are capable of 1800 seconds total operation. This conclusion is preliminary, and must be supported by additional experimental evidence.

Further evaluation of ablative chambers with varying carbon-cloth thicknesses and wrap angles is presently under way. Sufficiently long firing durations are planned to permit determination of char rates from the embedded thermocouples.

Regeneratively Cooled Throat Sections

Several 60-second firings have been made with this component to date. Pressure drop, heat load, and bulk temperature rise data were measured, using both water and monomethylhydrazine as the coolant. These preliminary results indicated satisfactory nozzle cooling, but any firm conclusions must be based on the results of the long-duration firings currently being conducted with the regenerative hardware.

Ablative Skirts

Several short-duration firings have been made with one skirt to check out the altitude facility. The full-scale skirt-evaluation-test series is presently under way, and no data are available for presentation at this time.

SUMMARY OF RESULTS TO DATE

The program thus far has resulted in: (1) the selection of monomethylhydrazine as the best compromise fuel with oxygen difluoride; (2) the selection, based on analysis, of an ablative-regenerative-ablative composite chamber, and of an injector design compatible with the selected cooled chamber; (3) the experimental verification of the ability of the injector to yield the required performance efficiency and maintain the structural integrity of the thrust chamber; and (4) an indication, based on experiment, that the selected thrust chamber configuration is probably capable of the required 1800 seconds firing duration, in either a continuous or a start-stop mode.

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